Background and Introduction:

A standardized spacecraft has long been a topic of discussion in the space community. Several vendors and customers have pursued the development of a standardized spacecraft to: a) achieve cost savings by minimizing non-recurring engineering for production vehicles, b) achieve operational savings by standardizing the tracking, telemetry and control (TT&C) functions, and c) improve schedule performance by being able to launch predictably and on need.

Today there are a wide variety of "standardized" spacecraft in use in the United States. However, the large quantity of "standardized" vehicles reveals the absence of a meaningful standard. Each vendor has a basic spacecraft bus, or two, or three, or more, from which they make modifications to provide a bus suitable for a particular customer or application. The market dynamic at work in the space industry has caused this to be the most efficient business model for vendors to use. The question posed by Detachment 12 of the Space and Missile Systems Center (SMC) is whether this is the most effective business model for industry to operate on and whether the Department of Defense (DOD) should modify the way it purchases small spacecraft in an effort to alter this market dynamic.

In reviewing the costs associated with developing, launching and operating spacecraft, one finds there are many items that contribute to an overall high cost for space systems. Launch costs have been a chronic problem and has received much attention within the industry and government. For small payloads in particular, the cost of a dedicated launch vehicle can easily become the largest single item in the program's budget. Similarly, payload costs are generally much larger than spacecraft bus costs. In these cases the spacecraft bus often undergoes repeated customization since these costs are not considered drivers for the program. The small incremental cost of spacecraft bus modification leads to a "get it perfect" mentality to meet the needs of the payload developer. Often the drive for perfection gets in the way of "good enough" where standardized interfaces are compromised through customization. The payload-centric nature of space systems and the apparently insatiable appetite for spacecraft modifications significantly impairs the emergence of a market leader for a standardized small spacecraft bus.

In essence, the current market forces in the small spacecraft market have established an anti-QWERTY effect. Most people are familiar with the layout of the standard English keyboard but many do not realize how this ubiquitous standard came about. Keyboards are not designed or intended to be efficient for the user's hand movements. Instead, keyboards are designed to prevent mechanical typewriters from jamming even though very few people use mechanical typewriters anymore. The standard gained so much market share that even as technology has advanced, the old standard still dominates the market. Unfortunately for the space business, with its payload-centric focus, there is no dominant design for spacecraft and customization remains in vogue. Although the Department of Defense has attempted to procure "standardized" spacecraft buses for programs like MightySat and WarFighter, follow-on payloads and mass production of these designs did not materialized and these programs became yet again additional design points in the overall mosaic of customized spacecraft designs.

If the payload-centric structure of the space marketplace is to blame to for constant customization of spacecraft, then one should ask: "What can be done to change the market forces to encourage rather than discourage spacecraft standardization?" A promising solution seems to lie in modifying the behavior of the government's purchasing practices for small spacecraft by concentrating less on the payloads and more on the space vehicle itself. There are examples from the air community that could provide useful insights for the space community. In the 1960s, the United States embarked on the development of several aircraft for single purpose missions or that were intended to carry a single payload type. Over time it became apparent that multi-purpose platforms provided a greater total capability even if the performance of individual missions suffered. This change coupled with effects-based targeting and the growing application of information technologies to munitions created great incentives to procure standardized air platforms over specialized platforms. While this experience has not yet transferred to the space program one can readily appreciate that such a development could take place.

How then can the DOD modify its procurement of small space systems to encourage the emergence of a standardized, multi-purpose spacecraft and a suite of interoperable payloads? The best way appears to be through economic incentives to counter-balance the current bias toward customization. Since the market forces of high launch costs and low bus costs have created the payload-centric market, the DOD should create a spacecraft-centric market by funding a separate acquisition activity for a small spacecraft bus. Taking delivery of the bus as a configuration item and purchasing payloads as a separate commodity would significantly restructure the military spacecraft market. This would be challenging is the initial stages since there would be a learning curve for military personnel as they would have a direct role in loading payloads to the vehicle and subsequently mating the completed spacecraft to the launch vehicle. However, the long-term operational responsiveness and financial savings of such an approach should outweigh the short-term costs and difficulties. The spacecraft bus acquisition agent should manage their launch vehicle funds to provide flexibility across launch vehicles and offer the payload community an "all expenses paid package" for payloads selected to be manifested. This structure should quickly re-order the payload community to adopt the standardized interfaces since it would be there ticket for a "free ride" into space. Such an approach would create a new center of gravity in the DOD small payload community around the standardize spacecraft bus and quickly capture the market share needed to set and enforce payload standards for a common interface. SMC Det 12 has proposed such a system in response to Combatant Commander interest regarding the potential for space cooperation with partner nations.

DOD Small Spacecraft Market Analysis:

The DOD small spacecraft market can be best understood by dividing it into two categories. The first category is Service or Agency funded spacecraft which are funded entirely by their sponsor. This category of spacecraft will be fairly immune to participating with a standardized spacecraft since it has the funding resources needed to pursue an independent path. In terms of total funding, these programs represent over half of DOD small space spending but they account for only 10% of small DOD payloads launched each year. The second category of spacecraft are those sponsored by the DOD Space Test Program (STP). Many of the programs start out in category one end up in category two when they apply for support from the DOD Space Experiments Review Board (SERB) and STP. STP is a centrally funded DOD program to assist payload developers with bus, launch and/or operational costs of flying their payload. Most DOD experiments end up going through STP on their way to space which is why STP represents roughly 90% of small DOD payloads launched while commanding less than 10% of the DOD small spacecraft funding. Notable examples of STP's reach include DARPA's Orbital Express and the OSD-sponsored Tactical Satellite 2 (TacSat-2) which have gone from stand-alone programs to benefiting from STP assistance.

Even though STP represents a minority share of DOD small spacecraft funding, it encompasses such a large share of the overall DOD small space programs to be the best subject of analysis to conduct a market analysis. For the purposes of this paper, STP data from fiscal years 1998 through 2002 will be used to describe the DOD small spacecraft market.

For the years 1998 through 2002, the SERB approved an average of 39 experiments each year for STP funding support. On average, 22 of these experiments were repeat applicants from a prior SERB and 17 were first time applicants. STP on average manifested eleven of these experiments and ten of them made it into space. Six of the experiments that were not manifested were not presented to a subsequent SERB. Based on these five years of data, it appears that STP flies only 25% of DOD payloads each year (i.e. 10 of 39). However, when one considers that most unsuccessful payloads continue to compete for funding in future years, the market analysis reveals much less elasticity. The market for DOD small spacecraft really centers on the 17 new payloads presented to the SERB each year. Of these, ten can expect to eventually succeed in getting STP funding sponsorship and get into space (typically within four SERB cycles), six will lose backing by their sponsor (due to either time delays, obsolescence or internal issues), and one will initially get STP support only to fall away prior to launch day. If one were to significantly increase funding for STP then the results should be: a) a shorter lag time between initial presentation to the SERB and launching into space, b) a smaller backlog of SERB approved payloads, and c) fewer payloads perishing due to obsolescence.

Summary of DOD SERB Payloads Average Annual Result for 1998-2002
Payloads Approved: 39
First-time Payloads: 17
Payloads Manifested: 11
Payloads Flown: 10

Table 1: SERB Payload Analysis

Assuming for a moment that one developed a standardized bus to carry five payloads at a time, then one would need to plan on two launches a year to dominate the DOD small spacecraft market by capturing over 50% of the payloads. Taking this approach should not only increase the volume of payloads flown and reduce the lag time to get to space but also create the market dominance needed to establish meaningful payload interface standards for the DOD market. This should significantly reshape the DOD small spacecraft market over the next decade. There are many ancillary benefits such as: a) creating a meaningful role for space operators in the small spacecraft market by flying a standardized spacecraft with operational crews, b) providing order and predictability to an otherwise apparently chaotic sector of DOD space programs (e.g. provide a prescribed path for spiral development), c) reducing launch processing times and increase launch flexibility and responsiveness, and d) allowing payload developers to concentrate on their core technical competencies without having to spend time and money on the bus and the operations concept.

Operational Concept:

Proceeding on with this conceptual approach, one should ask: What are the system-level requirements for such a spacecraft?" To best answer this question, one needs to have an idea for how the spacecraft is intended to be operated. SMC Det 12 proposed the following operational concept for a small spacecraft of this nature in order to derive system-level requirements for the bus.

Distinction between Bus and Payload Operations

For most spacecraft, the space segment is defined as the union of the bus and the payload(s). This has been typical of space system operations and design since the inception of the space age. Most space systems are designed around an end item payload which performs some needed function. In this context, the bus only exists to keep the payload functioning in space. The bus generally has no role or inherent value in itself. To re-order the small spacecraft business, a new operational concept is required that places emphasis on the bus as distinct from the payload operations.

Conjunction Operations

Spacecraft operations have historically, and continue in the present, to predominantly use rendezvous orbits. Rendezvous orbits refers to operations where a spacecraft is placed into a specific energy state defined by its semi-major axis, inclination, and argument of perigee. Use of these orbits only requires the spacecraft operator to station-keep in that position until the spacecraft is moved to another redezvous orbit or reaches its end-of-life. Traditional spacecraft operations are designed around the assumption that the spacecraft will be inserted into the desired rendezvous orbit and maintained there. For such systems, the main use of on-board propulsion is to station-keep by compensating for orbit perturbations although occasionally a spacecraft will carry additional fuel to allow it to maneuver from one orbit into another. Desirable rendezvous orbits are generally circular or critically inclined (i.e. 63.4°) to minimize orbit perturbations and the need to carry additional fuel for station-keeping. Conjunction orbits are those that are unstable and the orbit parameters will naturally precess. The continual relative changes of the orbit parameters cause such a spacecraft to cross paths, or conjunct, with the other objects in its altitude range. It is impractical to attempt standard station-keeping maneuvers for such a spacecraft since it would rapidly consume fuel. Unlike a rendezvous spacecraft, a conjunction spacecraft would use its fuel to maneuver a little every revolution (or day) to achieve mission parameters. An obvious application of such an orbit would be to overfly a specific geographic region at a predetermined time which could be arbitrarily chosen by a user on the ground. This possibility will be discussed in the orbital mechanics section of this paper.

Space Environment

Most people are aware of the hazards of operating in the space environment. Radiation exposure is generally a major concern for spacecraft design to ensure a spacecraft can endure the total radiation dose it will receive while in operation. Although it is possible to build radiation tolerant spacecraft at a modest increase over nominal spacecraft design, space mission planners generally avoid operations in the Van Allen Radiation Belts except to transit them on the way to higher orbit. Occasionally, it is impossible to completely avoid the radiation belts, as in the case of the South Atlantic Anomaly for low earth orbit spacecraft. However, it is safe to say this element of the space environment has had dramatic effect on the orbits chosen for space operations. The operations concept for conjunction spacecraft will require operations in the radiation belts with an above-average radiation tolerant bus.

Command and Control (C2)

Most spacecraft C2 concepts are designed around payload operations. Since they typically use rendezvous orbits, there is no point to commanding the spacecraft to do anything other than what gravity and its velocity vector will do on their own. Therefore, most space C2 issues center on mission priorities for who is using the primary payload. Spacecraft flying in conjunction orbits will open an entirely new field of options for C2. Now the spacecraft bus will need to be "piloted" to ensure the payload operators can achieve the desired time-over-target (TOT) needed for the new generation of payloads. The C2 concept now must focus on bus operations and leave the payload operations to function through the communications channels of the payload sponsor. Such a concept will allow terrestrial users to have a meaningful and direct influence on spacecraft operations and allow them to synchronize space operations to ground operations instead of just accepting a stable, predictable orbits for the spacecraft.

Mission Planning and Common Operational Perspective

Flying a conjunction spacecraft requires much greater spacemanship skills than normally associated with space operations. The art of spaceflight for such a vehicle will require new mission planning tools and a common operational perspective (COP) conducive to giving the spacecraft commander overall situational awareness of his craft and the hazards it might encounter. The current COP of plotting spacecraft ground tracks on a Mercatur-view map of the earth is insufficient for these types of operations. A new COP depicting "where" everything is in space is required. The "where" of space is not a 3-dimensional, time-driven view, but rather an understanding of the energy state of each object and whether two objects are likely to meet each other. This requires two views of space as an area of operations. First, one will need to plot the energy state of each object, most likely using altitude (on a logarithmic scale) and inclination. Second, one will need to depict the great circles each object travels in space to look for where they meet, or conjunct. The mission planning is similarly challenging in that one now will need to understand what orbit parameters the spacecraft is moving from and where it is going to. Maneuvering such a craft is an unbounded problem whereby an infinite combination of individual thruster burns could be used to achieve the desired flight parameters. The crew will be challenged to do this in the most fuel efficient and timely manner. New mission planning tools will be needed to accomplish the maneuvers while ensuring that no on-orbit collisions occur.

Crew Concept

The approach for conjunction operations described so far will certainly drive a new approach for the spacecraft crew. The approach taken for today's space operations are wholly inadequate since the space squadrons generally are managing a constellation of rendezvous systems where they are doing some orbit maintenance and mostly concerned with payload operations. Flying a single conjunction "tail number" will drive so many requirements onto the spacecraft crew that they will have to leave payload operations to the people responsible for those operations and concentrate on the spacecraft itself. Instrument Flight Rules (IFR) for space will be needed to dictate crew behavior and well as emergency procedures and requirements for positive control. New crew stations will be needed to perform the new functions of spacecraft navigation and manage the new instrumentation required to sense the environment around the spacecraft.

System Requirements:

With a good understanding of the operational concept, one can now derive the top-level system requirements for the new spacecraft proposed in this paper. The system concept is called XSM-1 in accordance with DOD Instruction 4120.15-L, "Model Designation of Military Aerospace Vehicles." XSM-1 denotes the system as an experimental scientific and calibration spacecraft. The following characteristics should allow XSM-1 to achieve its intended purposes:

Carry 150 Kg of Payload Weight

The analysis of the DOD small spacecraft market revealed that launching ten payloads a year would position one to dominate the market and be able to set and enforce payload and interface standards. There are significant variations in the size and weight of small spacecraft payloads. It is possible to give average values for size and weight but would not meaningful in this context since the largest "small payloads" would significantly skew the data. A more useful approach would be to use a median value or better yet, an approximate value for where one could capture 70% to 80% of the market. It would appear from reviewing the SERB list that 30 Kg is a reasonable value for the 70th percentile of small payloads (this is just the sensor package without the supporting bus functions). If one were to capture five at once for a total weight of 150 Kg, one could capture 80% of the market since not all of the payloads would weigh 30 Kg. For the five payload interfaces, three should be nadir-facing (i.e. towards the earth) and two zenith-facing (i.e. away from the earth).

Carry 30% of Total Weight in Fuel

On-orbit maneuver capability is typically described in terms of how many meters/second of velocity change the propulsion system can deliver. Using the rocket equation, this value can be converted into a percentage of how much the fuel weighs in comparison to the gross (or "wet") weight of the spacecraft. A typical spacecraft will get to orbit with 10% to 15% of its mass left as fuel for a 7-10 year operational life. This is usually acceptable since, for typical rendezvous operations, fuel is used much like an apogee kick stage where the propulsion system provides a "kick" to move the spacecraft into a transfer orbit then kicks again to "round-out" in the new orbit. The other main use is to conduct zero momentum burns to allow for de-saturation of the reaction wheels. This thinking significantly limits one's ability to grasp the opportunities for maneuvering from a conjunction platform and maneuver a little bit every day to achieve other mission parameters such as conjunctions with other space objects or moving from one TOT to another for the same geographic area. It appears a 30% mass fraction of fuel is sufficient to enable two years of spacecraft operations for a conjunction system. The anticipated fuel budget for XSM-1 is described in greater detail in the orbital mechanics section of this paper.

Radiation Tolerance

Most space payloads experience performance problems in a high radiation environment. A traditional radiation mitigation approach would be to shield sensitive components to ensure their functioning or survival in this environment. Shielding is generally undesirable since it adds significant weight to the spacecraft. Luckily the radiation belts are torroidally shaped which means the percentage of time spent in a high radiation environment is a small percentage of the overall orbit period. This means discontinuous payload operations may be a viable alternative where sensitive components could be powered-off during passes through the belts and lower any shielding requirements. The spacecraft bus is the one element that would have to operate normally in this environment and be radiation tolerant. Assuming the spacecraft will consume its fuel within two years, the on-orbit radiation tolerance for the bus would be approximately 100 Krads.

Interoperability

Interoperability is a key feature for such a system. The C2 system should be compatible with both the Air Force Satellite Control Network (AFSCN) and Unified S-Band (USB). The payload interfaces would need to be standardized both physically and electrically. The spacecraft bus would have to be qualified to survive the launch environment and meet the payload fairing requirements for all small launch vehicles that are likely to be used. The bus should be designed for standardization in every possible aspect to even consider internal interfaces to allow for component-level replacements to test new spacecraft components.

Operational Characteristics:

One should consider the operational characteristics of such a system to assess whether such an approach will make a meaningful improvement to future space operations. The principle issue here is whether there is a meaningful list of candidate payloads to provide useful information to users on the ground. The review of candidate small payloads reveals there are two main payload categories suitable for XSM-1. The first category is research-oriented programs that collect scientific data on the space environment or demonstrate new technologies. While an approach like XSM-1 is not ideal for these types of payloads, they could be enticed to cooperate given the financial incentives discussed. Considering these programs are usually low on funding anyway, they should be willing to participate even if the orbit parameters and radiation environment are undesirable. Some experimental payloads like the joint Army-Air

Force laser threat warning (LTWAR) detector could actually benefit from the on-demand geometries available from XSM-1. However, as a class, these payloads will have little interest to ground users until the technologies are proven and fielded in an operational form.

The second category is niche force enhancement systems for communications and surveillance. These systems would typically be payloads that provide real-time services while overhead. They would be persistent in terms of twice-a-day revisits over the area but not provide sustained 24-hour coverage since the idea is to fly XSM-1 by "tail number" and not field a constellation for continuous coverage. Leading contenders in this area are radio frequency (RF) payloads for communications relay or signals collection (such as blue force tracking). Optical payloads would probably not work well on the nadir surface since XSM-1 would not normally be used to fly in sun synchronous orbits. However, an imagery payload on the zenith surface would have opportunities to image all other space objects in XSM-1's altitude range over the course of nine months. In addition, space surveillance could be dramatically improved by allowing nearly on-demand conjunctions with other space objects to assess their status if the need arose.

Focusing one's attention on RF payloads should pay the great dividends to users on the ground since XSM-1 would enable new opportunities in terms of desired arrival times and observation geometries not currently available with rendezvous systems. The constantly changing ephemeris and unpredictable element set of XSM-1 could provide unwarned space operations for many RF payloads. Communication services, such as covert communications, could see dramatic improvements by being able to pre-schedule the spacecraft's time of arrival to fulfill communications needs with reduced likelihood of compromise to the user on the ground. Other active payload functions that are normally not considered for space platforms would suddenly be enabled by the flight profile and operational concept of XSM-1. This would finally allow the supported user to integrate space capabilities and effects into the overall plan for ground operations instead of having to live with pre-set spacecraft orbital mechanics associated with current space systems.

PeaceKeeper as a Space Launch Vehicle:

The Launch Vehicle (LV) selected to be used for the XSM-1 mission is the PeaceKeeper (PK) Space Launch Vehicle (SLV). Recently, Orbital Sciences Corporation was awarded the OSP-2 contract to provide refurbished PeaceKeeper missiles as a launch platform for the Rocket Systems Launch Program (RSLP). Some of the information in this section is based on data contained in the user's guide for that platform. To help analyze the PK in a SLV role, performance capability, fairing size and shape, and interface to launch site and vehicle will be discussed.

Performance Capability

Payload performance capability is a key decision point when selecting a LV. This was no exception for XSM-1. How much performance margin and boosted weight capability will the PK SLV have? To help analyze the performance capability of the PK SLV we will look at the loaded spacecraft weight and payload performance capability with performance margin.

XSM-1 is a small satellite in character with an approximate wet weight of 500 Kg or 1,100 lbs. The propellant carried on board the spacecraft is used entirely for mission operations. Thus, it is necessary for the LV to inject XSM-1 into the desired elliptical orbit without the spacecraft performing any maneuvers upon deployment. Table 2 below depicts the insertion accuracy of the PK SLV:

Error Type	Tolerance (Worst Case)	Error Source
Altitude (Insertion Apse)	+/- 10 nm (18.5 km)	Stage 4 motor performance uncertainty and guidance algorithm uncertainty.
Altitude (Non-insertion Apse)	+/- 50 nm (92.6 km)	Stage 4 motor performance and guidance algorithm uncertainty and navigation (INS) error.
Altitude (Mean)	+/- 30 nm (55.6 km)	Stage 4 motor performance and guidance algorithm uncertainty and navigation (INS) error.
Inclination	+/- 0.2 degrees	Guidance algorithm uncertainty and navigation (INS) error.

Table 2: PK Orbit Insertion Accuracy

Since we know the approximate weight of the spacecraft we can now determine the payload performance and performance margin by selecting a launch site. This is necessary since the LV will have different performance characteristics when launching from the different space ports due to parameters such as the launch azimuth and orbit inclination. For the purpose of keeping this simple, the two primary US launch bases are considered for calculating PK SLV performance margin.

The two figures below depict the performance capabilities of the PK SLV from Vandenberg, AFB and Kennedy Space Center (KSC) respectively. However, these performance curves provide the total mass above the standard, non-separating interface. The mass of any Payload Attach Fitting (PAF) or separation system is to be accounted for in the payload mass allocation.

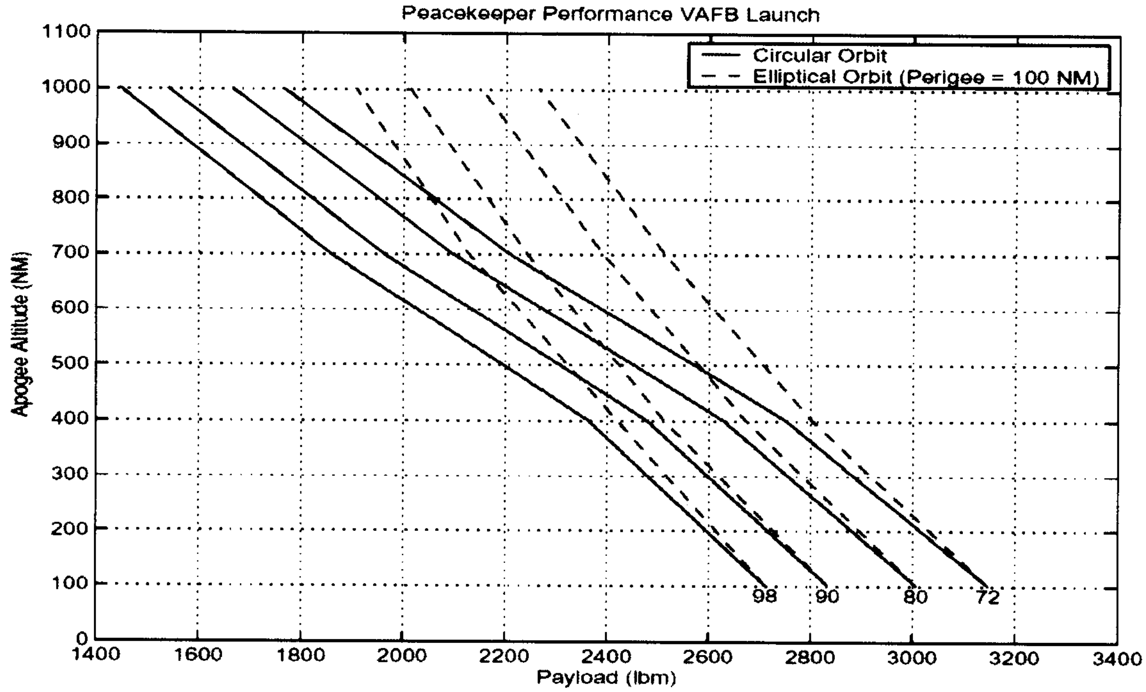

Figure 1: PK Lift Capability from VAFB

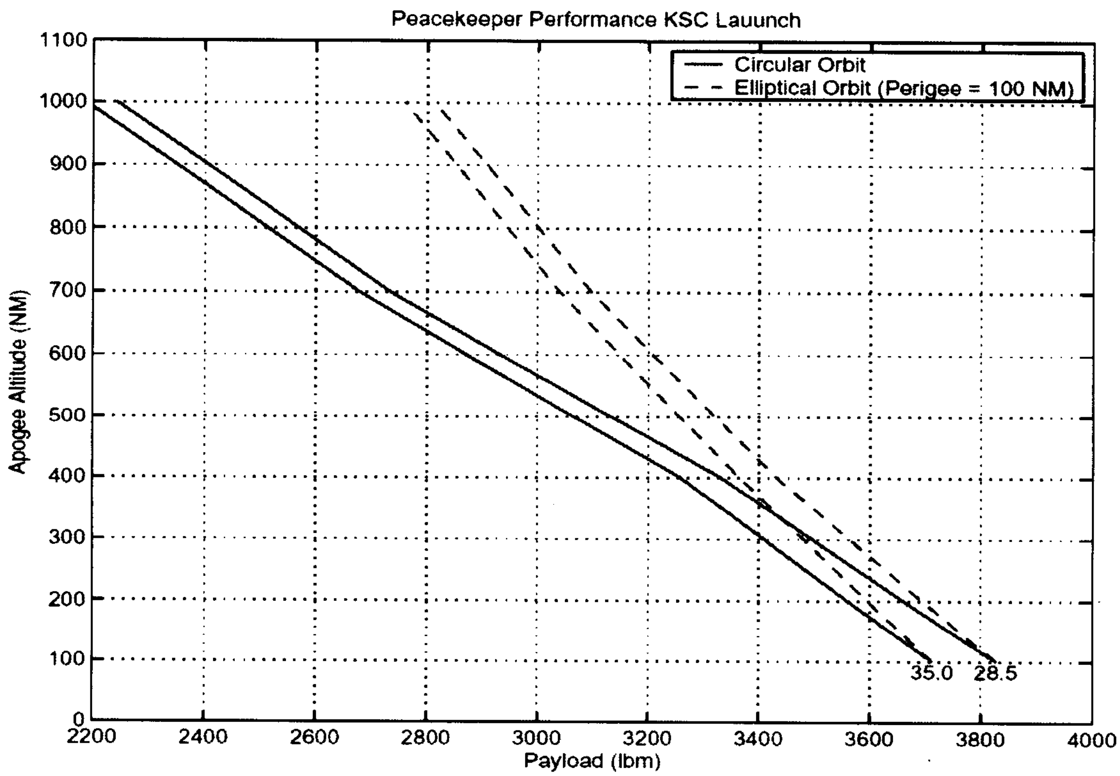

Figure 2: PK Lift Capability from KSC

These figures show the PK SLV performance for injecting into circular orbits. For XSM-1, the desired orbit is highly elliptical with an apogee of 6042 Km and perigee of 674 Km. At 500 Kg mass, it is on the performance edge of the PK SLV when launched from VAFB into an 85 degree inclined orbit. It is not possible to attain this orbit with a PK SLV from KSC.

Location	Nominal Payload Mass	Inclination	Performance Margin
VAFB	~2300	98.0	~1420
VAFB	~2650	72.0	~1770
KSC	~3200	35.0	~2320
KSC	~3300	28.5	~2420

Table 3: PK Lift Capability

Fairing Size and General Launch Vehicle Characteristics

Currently, the PK SLV is using the existing 92" diameter fairing, fixtures and handling procedures of the Taurus as shown below. The fairing is a bi-conic design made of graphite/epoxy face sheets with aluminum honeycomb core. The two halves of the fairing are structurally joined along their longitudinal interface.

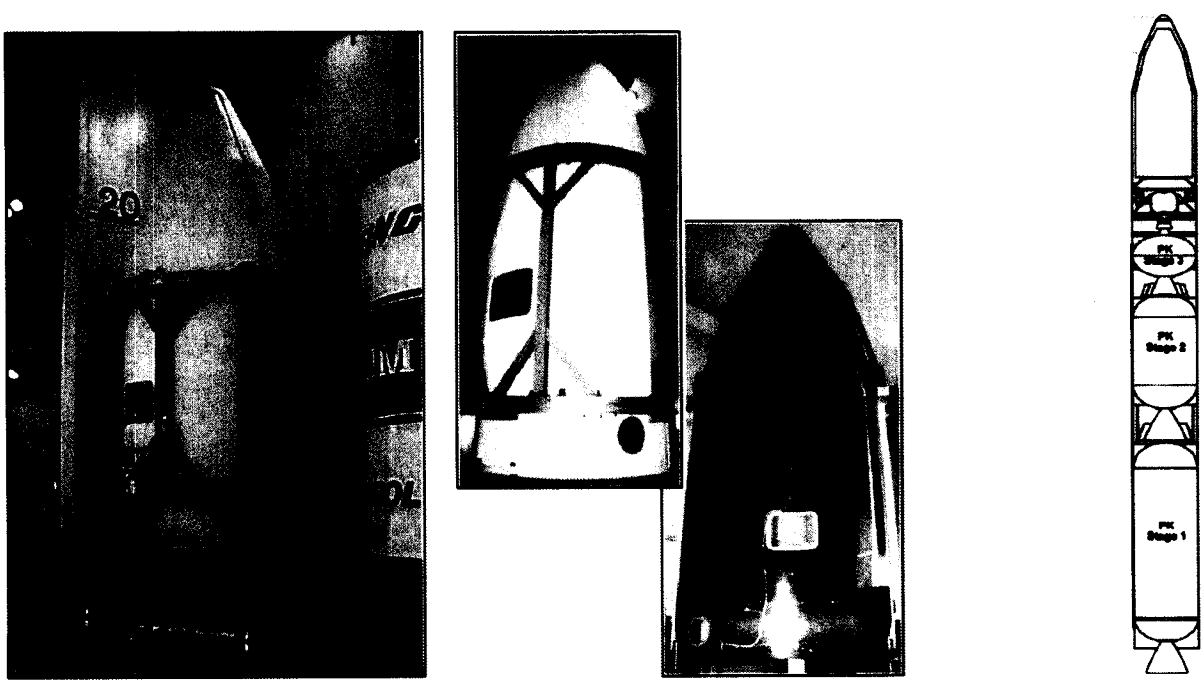

Figure 3: PK Fairings

The PK SLV shown in figure 3 is a four-stage, inertially guided, solid propellant ground launched vehicle. The first three stages consists of refurbished Government Furbished Equipment (GFE) PeaceKeeper stages 1, 2 and 3. The booster assemblies are used as provided by the Government, requiring no modification or additional components.

The fourth stage, Orion 38, provides the velocity needed for orbit insertion for the SLV. The Orion 38 is the same motor used with the Pegasus, Pegasus XL, Taurus and Taurus XL. This motor is currently in production with 37 successful flights and one test to date. This high level of reliability will next to guarantee availability of the motor when needed. The Orion 38 motor is shown in figure 4.

ORION 38

Ballistic Summary @ 70°F (294K)					
Burn Time (sec)	Max Pressure (psia)	Avg Max Vac Pressure (psia)	Avg Vac Thrust (lbf)	Vac Thrust (lbf)	Impulse (lbf-sec)
68.4	665	549	8,050	7,170	490,800
68.4	4.58	3.78	35.81	31.89	2,183.08
(sec)	(MPa)	(MPa)	(kN)	(kN)	(kN-sec)

Weight Summary		
Total Loaded Motor	1,976.9 lb	896.5 kg
Propellant	1,699.0 lb	770.5 kg
Total Inert	277.9 lb	126.0 kg
Burnout	252.0 lb	114.3 kg
Mass Fraction	0.86	0.86

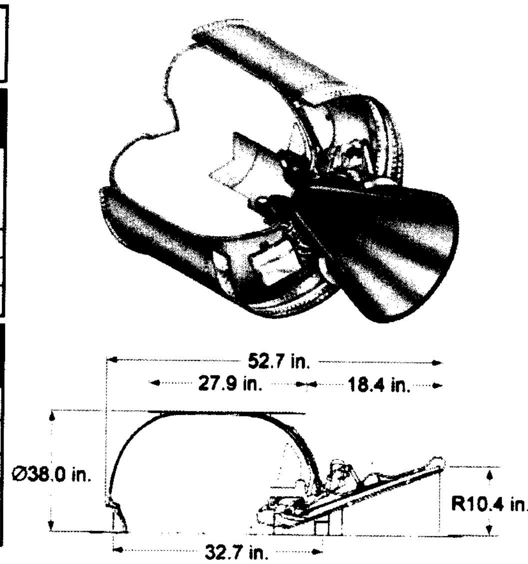

Figure 4: Orion 38 Stage

Three substructures are used to accommodate the Orion 38 Stage 4 motor and attach it to the Stage 1-3 PK booster assembly. These are, 1) a Payload Adapter Module (PAM) with 62.01-inch diameter payload interface, 2) the GCA +3/4 interstage and, 3) a 38-inch diameter motor adapter cone. These structures were adapted from similar Taurus hardware designs.

The 62.01-inch Payload Attach Fitting (PAF) is a scaled up Taurus 38-inch PAF. It is integrated with an adapter ring to create the PAM to which the fairing is mated. The ring interfaces with the forward joint of the GCA and the fairing frangible joint. The combination of the PAM and Fairing with the payload provides an integrated, contamination-controlled assembly that is transported from the payload integration facility to the launch site.

The Guidance Control Assembly (GCA) structure, is constructed from aluminum honeycomb substructure covered with graphite epoxy. The length of the structure is 41.93 inches. The GCA is constructed using six plies of graphite epoxy for both face sheets with a core thickness of 0.45 inches. Graphite epoxy is used as the face sheets for the bulkhead with an aluminum honeycomb core. The total thickness of the bulkhead is 2.0 inches. Each face sheet has eight plies of graphite epoxy sandwiching the aluminum honeycomb core. The PK-SLV avionics are mounted to the forward and aft sides of the GCA bulkhead. All bulkhead-mounted components are vibration isolated with the exception of the space integrated GPS/INS (SIGI) and the telemetry transmitters.

The PK-SLV avionics system has heritage to the Minotaur, OSP TLV, as well as Pegasus and Taurus designs. The avionics system design incorporates Orbital's flight-proven Modular Avionics Control Hardware (MACH). Standardized, function-specific modules are combined in stacks to meet vehicle-specific requirements. The functional modules from which the MACH stacks are created include power transfer, ordnance initiation, booster interface, communication, and telemetry processing. Orbital has designed, tested, and flown a variety of MACH modules, which provide an array of functional capability and flexibility. MACH has exhibited 100% reliability on all flights including OSP-1 Minotaur and TLV flights and several of Orbital's suborbital launch vehicles including Navy Theater Wide (NTW), Critical Measurements Program (CMP), and Storm.

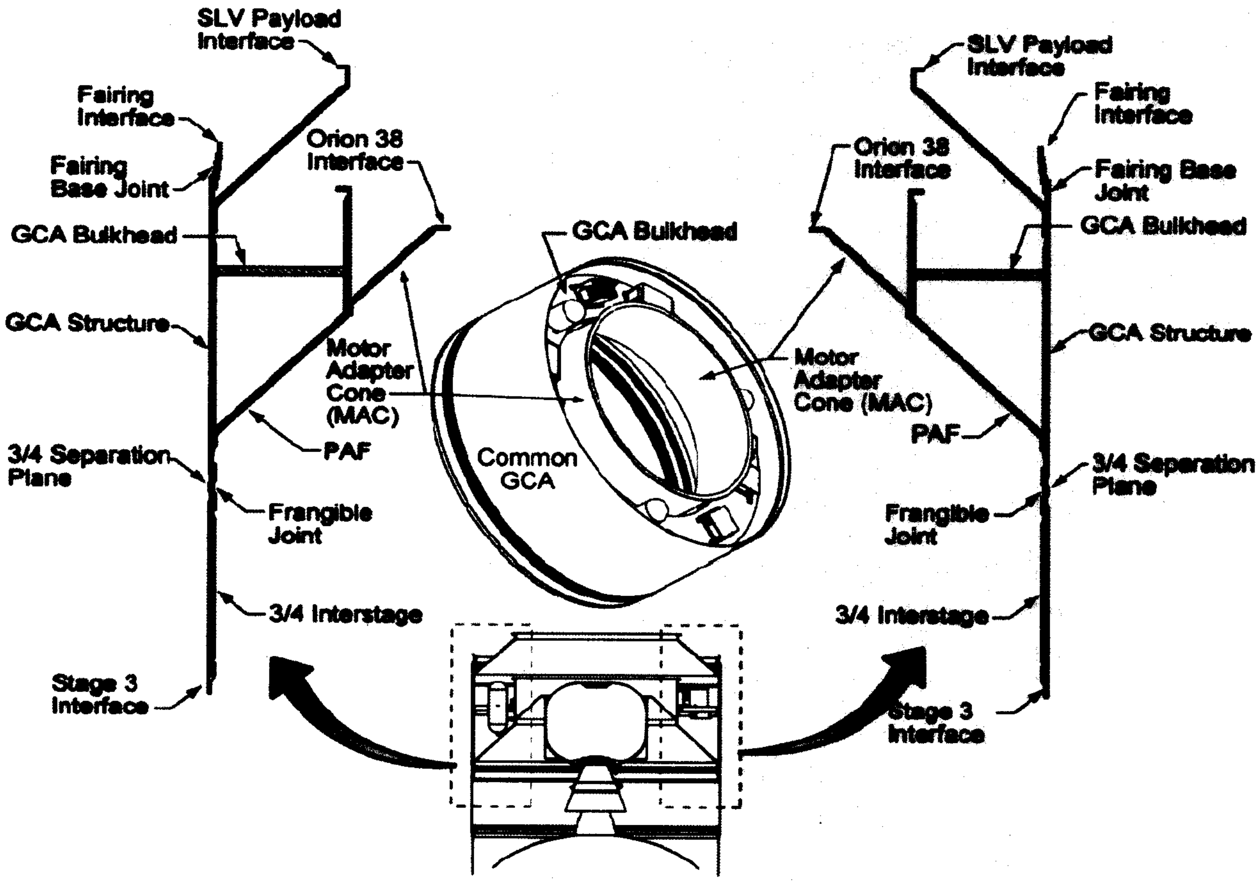

Figure 5: PK Guidance Control Assembly

The PK-SLV Attitude Control System (ACS) provides attitude control throughout boosted flight and coast phases. Stages 1, 2 and 3 utilize the PK Thrust Vector Control (TVC) systems. The PK Booster Control Module (PBCM) links the flight computer actuator commands to the individual Stage 1, 2, and 3 Thrust Vector Actuators (TVAs). Stage 4 utilizes the same TVC system used by the Pegasus, Taurus and Minotaur vehicles which combines single-nozzle electromechanical TVC for pitch and yaw control with a three-axis, cold-gas attitude control system resident in the avionics section providing roll control. Attitude control is achieved using a three-axis autopilot that employs Proportional-Integral-Derivative (PID) control. Stages 1, 2 and 3 fly a pre-programmed attitude profile based on trajectory design and optimization. Stage 4 uses a set of pre-programmed orbital parameters to place the vehicle on a trajectory toward the intended insertion apse. The extended coast between Stages 3 and 4 is used to orient the vehicle to the appropriate attitude for Stage 4 ignition based upon a set of pre-programmed orbital parameters and the measured performance of the first three stages. Stage 4 utilizes energy management to place the vehicle into the proper orbit. After the final boost phase, the three-axis cold-gas attitude control system is used to orient the vehicle for spacecraft separation, contamination and collision avoidance and downrange downlink maneuvers.

The PK-SLV telemetry subsystem provides real-time health and status data of the vehicle avionics system, as well as information regarding the position, performance and environment of the PK-SLV vehicle. This data may be used by range safety personnel to evaluate system performance. The PK-SLV baseline telemetry subsystem provides a number of dedicated payload discrete (bi-level) and analog telemetry monitors through dedicated channels in the SLV encoder. The PK-SLV telemetry system provides a baseline 1 Mbps data rate (both payload and PK-SLV telemetry). However, the output data rate is selectable from 2.441 kbps to 10 Mbps to allow flexibility in supporting evolving mission requirements, as limited by link margin and Bit Error Rate (BER) requirements. The telemetry subsystem nominally utilizes Pulse Code Modulation (PCM) with a RNRZ-L format. However other types of data formats, including NRZ-L, S, M, and Bi-phase may be implemented if required, accommodating launch range limitations.

Interfaces to Launch Site and Vehicle

The PK-SLV LSE is designed to be readily adaptable to varying launch site configurations with minimal unique infrastructure required. The LSE consists of transportable consoles that can be housed in various facility configurations depending on the launch site infrastructure. The LSE is composed of three primary functional elements: Launch Control, Vehicle Interface, and Telemetry Data Reduction.

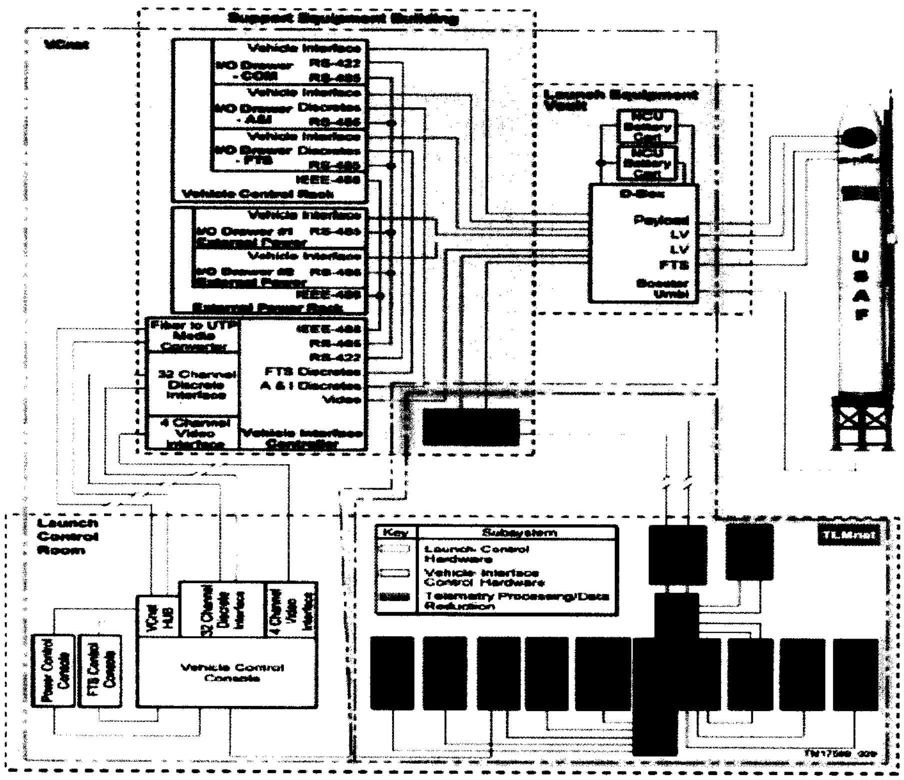

Figure 6: PK Launch Control

The Launch Control consoles are located in a Launch Control Room (LCR), or mobile launch equipment van depending on available launch site accommodations. The Vehicle Interface EGSE is located in a permanent structure, typically called a Support Equipment Building (SEB) or Launch Equipment Room (LER). Fiber optic connections from the Launch Control to the Vehicle Interface consoles are used for efficient, high bandwidth communications and eliminate the need for copper wire. The Vehicle Interface racks provide the junction from fiber optic cables to the copper cabling interfacing with the vehicle. The figure above depicts the functional block diagram of the LSE.

The LCR serves as the control center during the launch countdown. The number of personnel that can be accommodated are dependent on the launch site facilities. At a minimum, the LCR will accommodate personnel controlling the vehicle, two Range Safety representatives (ground and flight safety), and the Air Force Mission Manager. Interface to the payload through the PK-SLV payload umbilicals and landlines provide the capability for direct monitoring of payload functions.

XSM-1 Orbital Characteristics:

Based on the performance of the PK SLV, it is assumed XSM-1 will launch from Vandenberg Air Force Base, California. The Peacekeeper will launch in a southeast direction and dogleg to the determined orbit inclination. The Peacekeeper will be used to insert the spacecraft as close as possible into the baseline orbit to minimize any need for the spacecraft to use its fuel to make-up a shortfall in booster performance. To meet the objectives of XSM-1, a baseline orbit was selected which would allow for natural angular and temporal precession. The baseline orbit for a XSM-1 spacecraft is shown in table 2.

Orbit Parameter	Symbol	Value
Semi-major axis	a	9736 km
Eccentricity	e	0.2757
Inclination	i	85 deg

Table 2: XSM-1 Orbit Parameters

These parameters allow for an approximate apogee altitude of 6042 km and a perigee altitude of 674 km; assuming a value of 6378.14 km for the Earth's mean radius. Figure 7 below is a Satellite Took Kit (STK) depiction of the baseline orbit with a right ascension of the ascending node of 175 degrees and a mean anomaly of 320 degrees.

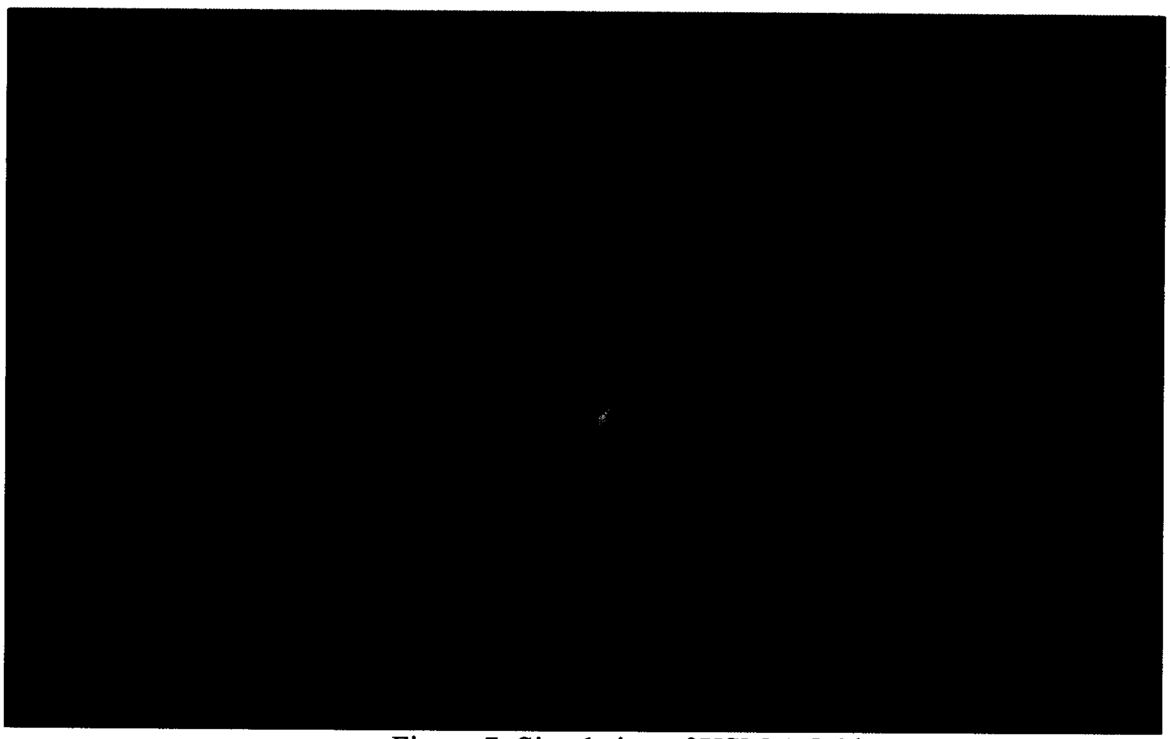

Figure 7: Simulation of XSM-1 Orbit

On-orbit Maneuvers

XSM-1 must be capable of performing various maneuvers to allow for a quick response to user tasking in performing conjunction operations. For the purposes of this paper, a quick response will be defined as allowing XSM-1 three days to reach the position in space needed to support the user. Three days was chosen based on the current three day planning cycle for the Air Tasking Order (ATO). This means XSM-1 operators need to be allowed to start maneuvering the vehicle soon after its "target" overflight window is nominated for inclusion in the Space Tasking Order (STO). The Air Operations Center (AOC) is expected to take two days to develop the ATO/STO with execution of the tasking order occurring on the third day. The operational lifespan of the XSM-1 spacecraft is 2 years; therefore it is imperative to be as fuel efficient as possible while performing any maneuver. Before defining the possible spacecraft maneuvers, the natural precession of the orbit will be discussed since they have a large impact on how tasking orders would be executed.

Orbital Precession Rates

This analysis assumes the Earth is not spherically symmetric with uniform density; therefore the Earth is not treated as a point mass. When looking closer at the Earth's mass distribution, it is apparent there is more mass distributed along the equator causing the Earth to bulge. The Earth's radius is approximately 22 km larger along the equator than through the poles. This oblateness is a principle contributor to the effect commonly known as the J2 effect. This J2 is unrelated to the theater commander's director of intelligence which goes by the moniker of the "J2." The orbital J2 is a constant describing the size of the bulge in the mathematical formulas used to model the oblate Earth. This effect changes the right ascension of the ascending node and moves the argument of perigee within the orbital plane.

Making use of the J2 effect, the perigee rotation rate can be calculated from equation 1:

$$\text{Perigee rotation rate} = 1.03237 * 10^{14} a^{-3.5} (4-5\sin^2 i)(1-e^2)^{-2} \qquad (1)$$

Based on this equation, the perigee rotation rate will be –1.28 deg/day. The major axis, or line of apsides, rotates opposite to the spacecraft's motion. The orbit will continue to rotate at this rate within the orbital plane, completing this cycle every 281.25 days (9 months).

This equatorial bulge causes the plane of the orbit to process resulting in a movement of the ascending node. This motion is westward for direct orbits (inclination < 90°). The nodal regression rate is a function of inclination and orbital altitude. To determine this rate, equation 2 is used:

$$\text{Nodal regression rate} = -2.06474 * 10^{14} a^{-3.5} (\cos i)(1-e^2)^{-2} \qquad (2)$$

Based on this equation, the nodal regression rate is 0.23 deg/day. This is the reason for the drift of the right ascension of the ascending node (RAAN). Essentially, the RAAN "walks back and forth" eastward and westward for a given cycle of 281.25 days. Simply put, you could say that the RAAN would go east for 140.6 days and west for 140.6 days. This would mean the spacecraft would flyover a range of longitudes of 32.2 deg. This estimate would be too high as the line of nodes will flip and the precession rate will stop and turn around. Using PCSOAP and Satellite Tool Kit (STK) as an alternative analysis tool, one finds the actual range of longitudes is bounded within approximately 14 degrees.

Since the period of this orbit is approximately 2.66 hours, the vehicle will make nine revolutions about the Earth per day. Based on this number of revolutions per day, there will be a natural temporal precession rate of –4.8 min/day. Simplistically speaking, the spacecraft will flyover a specific ground location 4.8 minutes earlier than the previous day.

The natural motion described above can be beneficial to conjunction operations and the XSM-1 mission. Should we require maneuvering of the spacecraft on-orbit, less fuel may be required to complete the maneuvers. Essentially, the spacecraft is "captive" over a finite bounded range of longitudes making it always on-call for theater support.

It was previously mentioned a user might request the spacecraft to fly over a particular location at a particular time; the question is how to accomplish this objective. Three possible maneuvers are discussed in detail below. These are timing maneuvers, longitudinal maneuvers, and argument of perigee rotation.

Timing Maneuvers

Within the context of this paper, a timing maneuver is defined as changing the time the spacecraft would routinely fly over a specific geographic region based on a time arbitrarily chosen by the user. As a practical matter, the maneuvers will control both the latitude and time of day the overflight occurs. To get the vehicle over the target, one would still need to consider the vehicle's longitude which is discussed in the next section.

The amount of fuel required to perform timing maneuvers will depend on the magnitude and direction of the change. What is beneficial about the baseline XSM-1 orbit is it allows for natural time changes. If the user would require the spacecraft to arrive over a location earlier than the previous day at any multiple of 4.8 minutes, fuel would not necessarily be required. For example, if the user required the vehicle to arrive over a particular region 24 minutes earlier, the vehicle would naturally be there within 5 days. However, should the user require the vehicle to arrive later than normal and if they needed the vehicle to be there within a few days, it would be necessary to apply a burn to accomplish this maneuver.

There are two obvious maneuvers that can change the time XSM-1 passes over a particular latitude. The first maneuver is to perform a zero-momentum burn to rotate the orbit within the orbit plane and directly reposition the vehicle to arrive at the desired time. This type of maneuver would not be fuel-efficient and is only noted in passing. The second maneuver is to change the orbit period to "walk" the vehicle north or south relative to a particular latitude. This would be done by firing the thrusters at apogee. Apogee burns are preferred since they significantly reduce the amount of fuel required. A reverse-thrust at apogee would lower the perigee altitude thereby decreasing the orbit period and causing XSM-1 to arrive at the desired latitude earlier in the day. A forward-thrust at apogee would raise the perigee thereby increasing the orbit period and causing XSM-1 to arrive later in the day. These maneuvers are intended to be conducted up to three days before the desired overflight to make maximum use of the natural precession of the orbit and to decrease the amount of fuel necessary.

In theory, the maximum amount of time adjustment XSM-1 would need to make is six hours since that is the halfway point between the ascending and descending node passes the spacecraft makes over any latitude in a particular region. The 2.66 hour orbit period causes the ascending and descending nodes to align with each other over nine different regions of the earth. This provides twice-a-day overflight of the region of interest and means XSM-1 is always within 12 hours of its next overflight. This also means that to schedule a future overflight, one can choose between the ascending or descending pass in order to save fuel. However, there is a limit to how much XSM-1 can shorten its orbit period in an effort to arrive over the desired latitude at an earlier time. If one were to attempt a maneuver to arrive 6 hours earlier, then the perigee would be lowered so far as to cause XSM-1 to impact the earth. As a consequence, Joint Munitions Effectiveness Manual (JMEM) for XSM-1 would need to indicate this as a system limitation so the planners would know not to attempt such a maneuver. In these cases, XSM-1 would actually delay the corresponding opposite pass by nine hours to achieve the intended effect.

Table 3 shows the amount of fuel required to perform these timing maneuvers. The assumption made in calculating these maneuvers is the burn is made 27 revolutions before the intended overflight since XSM-1 is provided three days warning of the request for support. From this table it is apparent that single burns propagated over many revolutions are a fuel-efficient means of maneuvering the spacecraft. It is also apparent that timing adjustments greater than three hours over a three day period quickly become fuel prohibitive. It would be a better use of XSM-1's fuel to delay the tasking by an extra day or two or provide an earlier notice of the emerging need for the support.

Delta Time	3 hrs earlier	1 hr earlier	same time	1 hr later	3 hrs later	6 hrs later	9 hrs later
Delta-v	94.1 m/s	11.6 m/s	27.6 m/s	65.8 m/s	139 m/s	241 m/s	335 m/s
Semi-major axis	9,512 km	9,694 km	9,785 km	9,874 km	10,054 km	10,319 km	10,581 km
Orbit period	2.56 hr	2.64 hr	2.68 hr	2.71 hr	2.79 hr	2.90 hr	3.01 hr

Table 3: Timing Maneuvers over Three Days

A closer examination of the "arrive at the same time" scenario reveals a recurring need to maneuver every three days to re-adjust the orbit. The 27.6 m/sec figure does not account for any compensation of longitudinal precession which will also occur during this time. As a rule of thumb, XSM-1 should not be used to overfly the same point on the earth for more than a week or two or else it will significantly decrease the life expectancy of the system. As a conjunction system, it is better suited to "fetch" new and emerging requirements than to try and repeatedly perform the same mission each day.

The preceding discussion assumed XSM-1 has a conventional attitude control system (ACS) which is used for both station-keeping and maneuver, such a system would typically have an Isp of 300 sec. Using the rocket equation in equation 3 reveals a total maneuver budget of 1,050 m/sec.

$$\text{delta velocity} = (g)(Isp)\ln(1/(1-m^*)) \qquad (3)$$

Given the relatively low thrust requirements to conduct XSM-1 maneuvers, an advanced ACS system such as a Xenon-ion propulsion system (XIPS) made by the Boeing Corp could be used. Such a system is capable of delivering an Isp in excess of 2,500 sec. This could increase the maneuver range of XSM-1 to 8,750 m/sec. Using an ACS system such as this could both extend the on-orbit life of XSM-1 and allow for eight times more maneuvers.

To illustrate the significance of the three day advance notice of maneuvers, table 4 shows the impact of trying to maneuver from one overflight time to the next within a single day. Only the values for later arrival are shown since the limit for earlier arrival is even further restricted by earth-impacting trajectories.

Delta Time	1 hr later	2 hrs later	3 hrs later	4 hrs later	5 hrs later	6 hrs later
Delta-v	190 m/s	290 m/s	380 m/s	460 m/s	540 m/s	610 m/s
Semi-major axis	10,027 km	10,293 km	10,555 km	10,814 km	11,070 km	11,323 km
Orbit period	2.78 hr	2.89 hr	3.00 hr	3.11 hr	3.22 hr	3.33 hr

Table 4: Timing Maneuvers in One Day

Longitudinal Maneuvers

Longitudinal maneuvers would follow along the same lines as a timing maneuver. In this case, the user would require the spacecraft to fly over a particular longitude that the vehicle would not normally fly over at a specific time. It is in this instance the natural nodal regression rate could become beneficial. Should the user need the vehicle over a location it would naturally regress to within a few days a burn may not be required.

The same maneuvers would be implemented to accomplish this goal as in the timing maneuver. The operator could raise or lower perigee to arrive over the given location. This maneuver is coupled with the timing maneuver. Timing maneuvers will also affect the longitude; therefore the XSM-1 flight crew would have to solve both problems simultaneously. A longitudinal maneuver should result in fuel savings except when you are trying to adjust time in the opposite direction from where you need to go with latitude.

Argument of Perigee Rotation

An argument of perigee rotation would be used to control the altitude of the spacecraft over a given location. This maneuver is listed as a possibility, but should be noted this maneuver should be used in special circumstances when there is no other way to satisfy the tasking order from the theater. For the purposes of the XSM-1 mission, one should be able to satisfy mission parameters by simply controlling the latitude, longitude, and time of arrival of the vehicle.

Argument of perigee rotations require a great deal of fuel to complete. Should this maneuver be required, the burn should take place at apogee so as to minimize the amount of fuel needed. Relatively small changes in argument of perigee rotation on the order of a few degrees could possibly be made with little impact to fuel consumption. However a rotation on the order of 90 degrees would have a huge impact to the amount of fuel remaining on board the vehicle. An argument of perigee rotation would not and could not take place on a regular basis simply because of the amount of fuel needed to complete the maneuver. This maneuver may not be able to be performed should this request come at the end of the life of the vehicle when little fuel would remain.

Fuel Budget

As mentioned, the goal of XSM-1 is to have a lifespan of two years on-orbit. To meet this goal, maneuvers will be implemented only after the flight crew has been able to mission plan the maneuver and assess the impact to the fuel budget. Normal flight rules should allow for normal maneuvers under a fuel threshold but require special consideration for high-fuel maneuver burns. Since the amount of fuel used will be dependent on the type of maneuver and how quickly the maneuver needs to be completed. The Commander of the 14th Air Force, Director of Space Forces or Joint Component Air Component Commander may need to intervene to make sure a judicious decision is made regarding fuel use. The proposed planning factor of three days is a reasonable planning consideration but would need to be worked within the context of the need of the theater users. In cases where the user is able to allow for additional time, it is possible to minimize fuel expenditure or use virtually no fuel at all.

Radiation Analysis

The radiation environment of space can lead to several problems for spacecraft including heating on exposed surfaces and degradation or damage to surfaces and electronic components. Electronics in space need to operate at about normal room temperature (20^{o} C or 68^{o} F). As previously stated, the electronic components aboard the spacecraft, as well as the spacecraft itself, will have to be shielded, or hardened, to handle the environment.

Perhaps the most dangerous aspect of the space environment is the pervasive influence of charged particles. The three primary sources for these particles are solar winds and flares, galactic cosmic rays (GCRs) and the Van Allen radiation belts.

The sun puts out a stream of charged particles as part of solar wind. During intense solar flares, the number of particles ejected can increase dramatically. GCRs are particles similar to those found in solar wind or solar flares, but they originate outside of the solar system. As the solar wind interacts with Earth's magnetic field, some high-energy particles get trapped and concentrated between field lines. These areas of concentration are the Van Allen radiation belts and they occupy a region from pole to pole around Earth.

As was previously mentioned, conjunction spacecraft like XSM-1 will be required to operate within the Van Allen radiation belts. An analysis using the CREME96 models was performed using the XSM-1 baseline orbit parameters. It was calculated the spacecraft would be subjected to an average radiation dose of 1.634E-3 rad (Si)/s or 51.55 krad (Si)/year not including any electrons which are not a part of CREME96 models. Over the course of 2 years, the spacecraft will be subjected to 1.031E5 rad/yr.

Numerous studies have been done on materials and their ability to withstand large doses of radiation. At the levels described above for XSM-1, there should not be a problem hardening the vehicle to withstand these levels and traveling through the Van Allen radiation belts should not pose a problem to mission success.